Space Station Freedom Electrical Performance Model

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The baseline Space Station Freedom electric power system (EPS) employs photovoltaic (PV) arrays and nickel/hydrogen (NiH_2) batteries to supply power to housekeeping and user electrical loads via a direct current (dc) distribution system. The EPS was originally designed for an operating life of 30 years through orbital replacement of components. As the design and development of the EPS continues, accurate EPS performance predictions are needed to assess design options, operating scenarios and resource allocations.

To meet these needs, NASA Lewis Research Center (LeRC) has, over a 10 year period, developed SPACE, (Station Power Analysis for Capability Evaluation), a computer code to predict EPS performance. SPACE was developed as a detailed *source-driven* model to predict the amount of power that the EPS can produce, throughout its life, under a variety of orbital conditions and on-orbit configurations. SPACE can be used alternatively in a *load-driven* mode, where the ability of the EPS to meet a time-varying, distributed load profile is assessed. Most satellite EPS designs are characterized by large capability margins, a small number of fixed loads and short design lifetimes. Freedom's EPS, however, will be operated as a utility at levels close to its maximum capability. A variety of loads must be accommodated and precise load profiles are not known a priori. Therefore, greater modeling detail is required in SPACE than in previous computer models. This detail is also needed because EPS performance requirements are verified by computer model predictions since size and scope of the EPS prevents a complete end-to-end system test to verify the EPS capability.

This paper describes SPACE, its functionality and capabilities. A companion paper presents computational results for some recent SPACE design analysis cases (Kerslake 1993). These include load driven assessments during Space Shuttle Orbiter rendezvous with an early build phase of Freedom, and an assessment of the capability of the EPS after 5 years of on-orbit operation.

Model Overview

SPACE was designed to predict the highest power level which the EPS can produce and sustain throughout the sunlight and eclipse portions of a specific orbit. The EPS consists of silicon solar arrays, NiH_2 batteries, a single phase thermal control system (TCS), and associated power conversion, control and distribution hardware. The EPS architecture is channelized and consists of two independent power channels for each PV power module (PVM) on-orbit. At the permanently manned capability (PMC) phase of Freedom, three PV modules are on-orbit. Fig 1 shows a simplified block diagram of a single power channel.
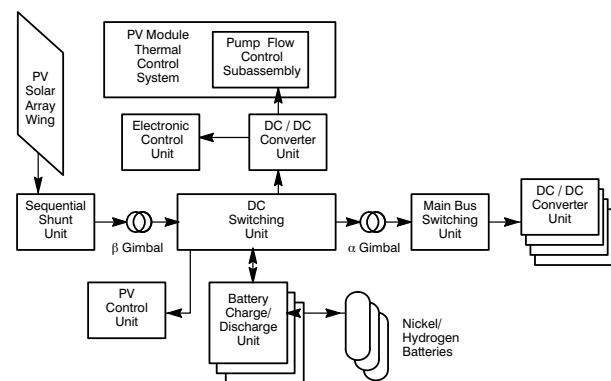
Fig 1. EPS Architecture Block Diagram

SPACE contains modules which model various aspects of the EPS and are linked together by an overall control program. The important modules are described in detail in the following sections.

Orbit Mechanics

Freedom will fly in a near-circular low earth orbit (LEO). The orbit mechanics section of SPACE determines needed orbit parameters such as sunlight and eclipse times, solar flux, and the solar beta (ß) angle (the angle between the orbit plane and the earth-sun line). Model inputs include orbit altitude, orbit inclination and flight date. In addition, the orbit to be analyzed can be controlled by specifying either an orbit precession angle or a desired solar ß angle, and the remaining orbit parameters will be determined by the code.

The orbit mechanics code determines the orientation of the Freedom core body and solar arrays relative to the earth and the sun. The core body is normally maintained in a near constant orientation relative to the earth, while α and ß gimbals articulate to orient the solar arrays toward the sun. The α gimbal rotates 360°, once per orbit, to compensate for the station revolving around the earth. The ß gimbal rotates ±52° in order to compensate for the variation of the solar ß angle. In this normal operating mode, the ß gimbal is also adjusted to prevent adjacent array shadowing. Also, angular errors induced by structural distortions are compensated for in the computation of α and ß gimbal settings.

SPACE allows the user to specify both Freedom's core body orientation relative to the earth and the α and ß gimbal settings. Gimbal rotational ranges can be restricted to simulate locked or partial rotation. Gimbal errors can be applied to deliberately off-point the arrays, and the gimbals' rotational speeds can be limited. The code includes an optimal ß gimbal positioning algorithm that optimizes array pointing in cases where the α gimbal is not fully tracking the sun.

Once the vehicle and solar array orientations relative to the earth and sun are determined, SPACE computes the solar, albedo, and planetary environmental heating rates on the arrays, radiator, and other surfaces. In addition, view factors between adjacent solar arrays are determined. This information is used by the solar array and PVM TCS models described below.

Solar Array Model

The solar array wing performance is determined by scaling the performance of an individual solar cell. The cell current-voltage curve is modeled using the single diode *Hughes model* (Rauschenbach, 1976) requiring four input parameters based on cell test data: 1) short-circuit current, 2) open-circuit voltage, 3) maximum power point current, and 4) maximum power point voltage. These data, for 7 different cell current grades, are corrected for environmental degradation effects (ionizing radiation, micrometeoroid/ space debris, contamination, plasma, ultraviolet irradiation, and thermal cycling), cell-to-cell current mismatch, solar vector off-pointing angle (due to array flatness, structural alignment, and solar tracking errors), and operating temperature. All correction factors are based on test data.

Transient temperatures are calculated for the front and back of the two solar array blankets which make up each wing. All cells on a blanket are assumed to operate at the same temperature although array circuit-by-circuit temperature calculations can be performed. Heat loads from solar insolation, albedo, earth infrared radiation, and adjacent solar arrays are factored into the array energy balance in addition to cell electric power output and resistive heating. For each blanket, cell operating temperature and cell power output are iteratively determined throughout the orbit.

Solar array circuit performance is obtained by scaling cell performance to represent 400 series-connected cells. Inter-cell and circuit conductor voltage drops are calculated and cell operating voltage interactively adjusted to satisfy

the voltage set point of the sequential shunt unit (SSU) that controls array voltage. Failed bypass diodes (one per eight cells) and solar cell shadowing are factored into this voltage iteration.

The SSU controls its output voltage by actively shunting and un-shunting the 82 solar array circuits which are connected in parallel on the two wing blankets. Since circuits on separate blankets operate in parallel, the wing voltage set point is 8 volts less than the orbit noon maximum power voltage (V_{mp}) of the higher temperature, lower voltage blanket. This is done to preclude operating the solar array at voltages above V_{mp} where EPS load following capability is diminished. Finally, wing power output is obtained by summing all circuit currents and multiplying by the wing operating voltage.

Battery Model

The batteries chosen for the EPS consist of individual pressure vessel NiH_2 cells. Each battery consists of two battery orbital replacement units (ORUs), each containing 38 cells connected in series. (Each battery has 76 cells in series.) A battery ORU has a nameplate capacity of 81 A-hr.

The battery model in SPACE is empirical in nature and calculates performance by extrapolating from cycle test data from NiH_2 cells similar to Freedom's design. SPACE contains two versions of the battery model. Initially, a version was developed by LeRC based on cycle data from a 65 A-hr cell on life test at LeRC. The second version, recently added, is based on data and algorithms provided by Space Systems/Loral, the EPS battery contractor (Wilson 1992). This version is based on cycle performance data from 72 A-hr boilerplate cells.

During discharge, the model determines voltage, current, and heat generated based on the power profile applied to the battery by the battery charge/discharge unit (BCDU). During charge, the model calculates the same parameters as for discharge, but based on a defined charge current profile envelope, instead of a power profile. Charge profiles can be defined in terms of current versus state-of-charge (SOC) or current versus time. For Freedom, the planned charge envelope consists of a constant current charge to 96% SOC with a linear taper to 2 A at the end of insolation. In charge, the model also accounts for coulombic efficiency determined as a function of SOC, charge rate, and battery lifetime.

Whereas the solar array model degrades the array performance using on-orbit exposure times, the battery performance degradation model is more complex. The SPACE battery life model determines a battery life *fraction* used for the given analysis, derived from the previous history of battery operation, input in terms of time spent at a particular depth-of-discharge (DOD). This life model is based on a number of completed and on-going LEO cycle tests (Wilson 1992). If the battery reaches end-of-life (EOL) before the point in time for which the analysis is being performed, the model automatically replaces the batteries on that particular PV module.

Distribution and Control

The power distribution model determines the current and voltage at each point in the system, as well as the power losses in each of the ORUs in the EPS. Each ORU is modeled appropriately for its function, with losses accounting for parasitic power usage, converter efficiencies, resistive losses and diode voltage drops. For example, the BCDU model includes: parasitic power consumption, efficiency curves for the charge and discharge converters, and resistive losses through internal devices and circuits such as the fault isolator.

In the eclipse, the power produced by the batteries is converted by the BCDU from low voltage (120 V nominal) battery power to high voltage (160 V nominal) bus power. The dc switching unit (DCSU) combines the power from the batteries on that channel and distributes power to: 1) the inboard power management and

distribution (PMAD) system which distributes power to the loads, 2) PV module loads, which power the ß gimbal motor and the PVM TCS pump, and 3) the SSU to provide control power. Losses are accounted for all the ORUs including the various cable runs between ORUs.

The PMAD system consists of main bus switching units (MBSUs), dc/dc converter units (DDCUs), and associated cabling. Power is distributed evenly among the DDCUs connected to each channel, and resistive losses are accounted for in the cables and the MBSUs as well as converter losses in the DDCUs.

In the sunlight, power is provided by the arrays via the SSU. The DCSU then distributes the power to: 1) the inboard PMAD system, 2) the BCDUs to charge the batteries, and 3) PV module loads. The BCDU now operates in a charging mode and so its losses are accounted for differently than during eclipse. Losses in other system components are modeled the same as during eclipse.

The EPS control system is modeled as it is designed to function on orbit. In the sunlight, the SSU is given an operating voltage setpoint based on the age and flight mode of the solar array. If the battery charge current can be sustained at the desired charge profile, the SSU will maintain its output voltage at the setpoint. If the power to the loads is high enough that the battery charge current cannot be maintained at the desired rate, the control of the bus voltage transfers to the BCDU which adjusts the battery charge current to maintain the bus voltage to a level somewhat below the SSU setpoint.

During eclipse, the SSU is not providing power, so the BCDU regulates the bus voltage to its setpoint.

Thermal Model

Each PVM has a dedicated single-phase ammonia thermal control system (PVM TCS) to collect, transport, and reject waste heat and to maintain the NiH_2 batteries within an operating temperature range between 32 to 68 °F. The PVM TCS includes a deployable dual loop

radiator, two pump and flow control subassemblies, and a heat removal fluid circuit including single-tubed radiant finned heat exchangers which allow easy replacement of the electrical ORUs.

The PVM TCS components are modeled as a resistor-capacitor network. The transient temperature distribution of the components is computed using standard finite difference, convection, and radiation heat transfer algorithms. The boundary conditions for the PV TCS radiator and other ORU thermal models consist of the environmental heating rates as determined by the orbit mechanics code and the heat dissipation rates determined from the battery model and the power distribution models. The PVM TCS code models the radiator mass flow rate by-pass valve, battery heaters, and radiator anti-freeze algorithms to properly simulate the on-orbit control system. The PVM TCS model can also predict temperatures of passively cooled components in the event that the active fluid control system should fail.

Optimization

Since the goal of SPACE is to predict the highest power level which the EPS can produce and sustain throughout the sunlight and eclipse portions of an orbit, an optimization must be done to maximize EPS capability. SPACE begins in the eclipse portion of the orbit selecting a power level to be provided. The model steps through the eclipse time, calculating the state of the EPS, including battery SOC, which results from the selected power level. The code then determines, through an iterative solution, the highest sunlight operating power level which will allow the batteries to return to full charge during the sunlight. In both the sun and eclipse portions of the model, the state of the EPS is checked at each time step to assure no hardware limits are exceeded.

When eclipse and sunlight power levels are determined, the code must determine if they are the *optimum* power levels. The code applies the following rules to determine the optimum power

levels:

    1) If the maximum battery DOD in the orbit is greater than the user-specified maximum, the eclipse power is too high and is lowered.

    2) If the sunlight power is less than the eclipse power, the power levels are not optimum, and the eclipse power is lowered.

    3) If the sunlight power is greater than the eclipse power, but the battery DOD is less than the user-specified maximum, the eclipse power is too low and is raised.

    4) If the sun power is greater than the eclipse power, and the battery DOD is equal to the user-specified maximum then the power levels are optimum. This is referred to as a *battery limited* condition.

    5) If the sun and eclipse powers are equal, then the power levels are optimum. This is referred to as an *array limited* condition.

If one of the first three conditions is true, the model proceeds to guess a new eclipse power level and the process is repeated. A combination of linear interpolation and bisection methods is used to converge to a solution quickly. The entire optimization procedure is performed for each power channel independently to assure the maximum power level from each channel is predicted.

Usage and Operation

A SPACE analysis is controlled by a single input file, in which the user can specify various control parameters. These include the assembly flight to be analyzed, the flight mode and pointing definitions, the orbit definition and the maximum allowed battery DOD. In addition to analyzing a single orbit, SPACE can be used in a variety of other manners, including: analysis of each flight throughout an entire assembly sequence, analysis of a series of consecutive orbits where array pointing is changing (i.e. a Shuttle docking maneuver), or analysis of the EPS capability through an entire year taking into account the changing orbit conditions and aging of EPS components.

Additional input parameters are used to control program output including the number and types of plots to be produced and the types of output files to be produced. Output plots can be displayed on the screen or sent to a PostScript® printer for printing. Fig. 2 contains a sample plot which shows the solar beta angle and eclipse time as they vary throughout a typical year. Another sample plot is shown in Fig. 3, which displays the EPS capability through a year for the permanently manned phase of Freedom.
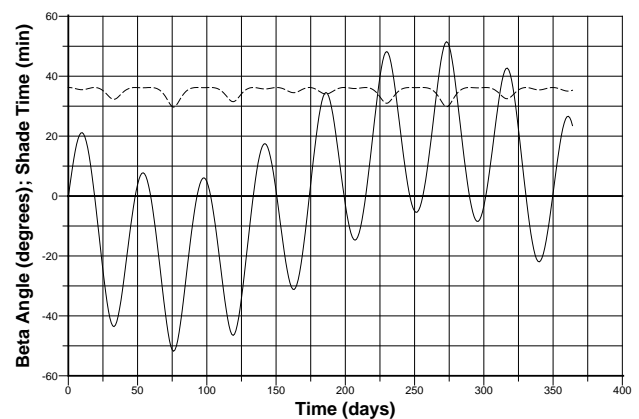
Fig 2. ß Angle and Eclipse Time Through a Year
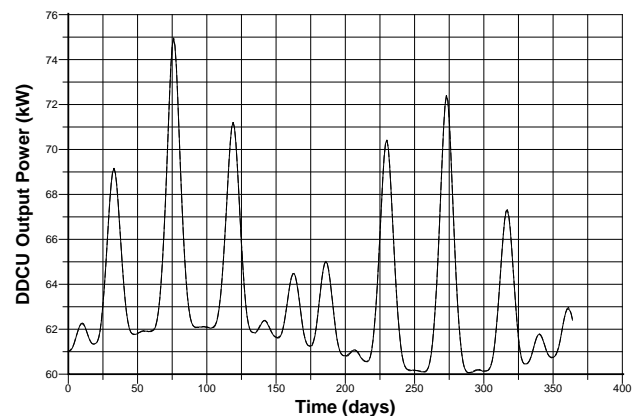
Fig 3. EPS Capability Through a Year for the PMC Phase of Space Station Freedom

In addition to an input file, SPACE references a large number of data files which contain information on the performance of

ORUs and components. A configuration controlled set of data files is maintained which contain the latest data for the EPS components. These are updated regularly as new data become available. The user can modify any of the data files as needed to perform a particular analysis.

Load-Driven Operation

In the Space Station Freedom program, Integrated Operations Scenarios are used to measure the performance of Freedom's design against sample mission scenarios. SPACE has been used in the load-driven mode to assess the EPS design against these mission scenarios. Three key SPACE inputs are needed for these assessments. First, the timeline for each electrical load on Freedom is defined throughout the scenario. Second, the changes in vehicle attitude, such as from the normal flight mode to the attitude used for reboost, are specified. Third, the periods when the solar arrays are locked in position, as occurs during the approach and departure of the Space Shuttle, are noted.

With these inputs SPACE determines the battery DOD throughout the mission scenario. This parameter is critical in gauging the feasibility of a specific mission scenario, as the batteries must never become fully discharged. Battery DOD is also the primary factor in determining battery life. SPACE also computes the electrical load that must be shed to maintain the EPS components within their hardware performance envelopes. This load shedding estimate is useful in identifying portions of the sample scenarios that must be revised to stay within the capabilities of the EPS.

When assessing these scenarios, the load-driven version of SPACE has one key degree of freedom relative to the source-driven version. Namely, the load-driven version does not require the batteries to be fully recharged at the end of each sun period. In an assessment including an Orbiter arrival, for example, with the array orientation dictated by plume load factors, the energy from the solar array may not be enough to completely recharge the batteries

for one or two orbits.

The load-driven version of SPACE shares the component models and component performance parameters with the source-driven version of SPACE. Greater than 80% of the source code is common between the two versions. The load-driven version was verified using the EPS capability calculated by the source-driven version as an input. For the same orbit conditions, both models calculate virtually identical values for parameters such as solar cell average temperature, battery DOD, and average bus voltage.

As the Space Station Freedom program progresses from the design phase to operations, the use of the load-driven version of SPACE will increase. For example, the assessments of sample mission scenarios will continue as users strive to utilize the full capability of the EPS.

References

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Acknowledgements

codes, some of which were incorporated in whole or in part into SPACE.